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Swarm RPO and Docking Simulation on a 3DOF Air Bearing Platform

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Abstract

With the emergence of the space servicing sector, along with the return of manned missions beyond low earth orbit, there is a need for quick, efficient, and most of all, safe Rendezvous and Proximity Operations (RPO). More than that, the next big step forward is manufacturing in space, which will require large swarms of spacecraft cooperating in close proximity to each other, all subjected to the same laws of orbital mechanics. Methods for swarm RPO safety are being developed but have not yet been tested in space. The most promising type of swarm RPO safety utilizes real-time GNC algorithms coupled with a variety of sensor inputs giving the position, velocity, and pose of all satellites in the swarm, to constantly update the relative-motion orbits of all the elements in the swarm, while propagating these orbits forward in time to prevent conjunctions. The University of Southern California's Space Engineering Research Center (SERC) is developing an in-house manufactured 3-DOF Air Bearing Platform (ABP), which has the ability to simulate the frictionless environment of space in a single plane. Real-time algorithms for swarm operations are planned to be tested with representative floating platforms. In preparation for platform operations, algorithms developed for swarm RPO were tested in a software simulation of the ABP hardware. Software based verification on the simulated ABP platform allowed for testing of various sensor configurations on the swarm elements, as different trajectories and approaches will have different range variations and rotation rates, all of which determine how well a given sensor can identify and update the position of the target spacecraft, and the other spacecraft in the swarm. The results of the initial simulations will be presented in this paper.

Keywords: Satellite, Swarm, Rendezvous, Testbed, Autonomous

Nomenclature

CLIENT .... Satellite or Platform to be Serviced

SERVICER ... Satellite or Platform that provides Service

SWARM ... Group of two or more spacecraft cooperating towards a common task or goal

Acronyms/Abbreviations

ABP .... Air Bearing Platform

C-W .... Clohessy-Wiltshire

CAD .... Computer Aided Design

CONFERS .. Consortium for Execution of Rendezvous and Servicing Operations

FOV .... Field of View

GEO .... Geostationary Earth Orbit

GNC .... Guidance Navigation and Control

ISS .... International Space Station

LEO .... Low Earth Orbit

LVLH .... Local Vertical Local Horizontal

OOS .... On Orbit Servicing

RPO .... Rendezvous and Proximity Operations

SERC .... Space Engineering Research Center

USC .... University of Southern California

1 Introduction

With the emergence of the space servicing sector, along with the return of manned missions beyond low earth orbit, there is a need for quick, efficient, and most of all, safe Rendezvous and Proximity Operations (RPO). More than that, the next big step forward is manufacturing in space, which will require large swarms of spacecraft cooperating in close proximity to each other, all subjected to the same laws of orbital mechanics. Methods for swarm RPO safety are being developed but have not yet been tested in space. The most promising type of swarm RPO safety utilizes real-time GNC algorithms coupled with a variety of sensor inputs giving the position, velocity, and pose of all satellites in the swarm, to constantly update the relative-motion orbits of all the elements in the swarm, while propagating these orbits forward in time to prevent conjunctions.

Through the course of research on historical RPO operations [1] and safety criteria for RPO and On-Orbit Servicing (OOS) [2,3] at the University of Southern California's (USC) Space Engineering Research Center (SERC), a set of informational databases on RPO and OOS were developed. These were used to develop trajectories for swarm RPO, enabling a variety of sensor inputs to obtain and update position, velocity, and pose for all the spacecraft in the swarm in real-time.

Using the capabilities being developed at the SERC, these methods were evaluated and tested on a software simulation platform to determine their effectiveness for swarm operations.

2 Background

In terms of orbital mechanics, RPO is the process of a spacecraft (Servicer) approaching and matching the orbit of another spacecraft (Client) [4]. RPO has been performed successfully since the 1960s, first demonstrated during the Gemini missions [5]. Current RPO methods still focus only on one-to-one operations [6–16]; that is, a single Servicer and a single Client. To foster an environment open to advanced multi-platform operations (i.e. in-space manufacturing or assembly), there first needs to be a framework in place to allow multiple Servicer's to operate on a single client, or even multiple Servicer's to multiple clients in the same vicinity.

Swarm RPO will enable multiple new capabilities on orbit, where two next-generation operations may be adaptive formation flying and satellite aggregation. Adaptive formation flying is the process of multiple spacecraft operating in relative motion orbits, within a few kilometers of each other, working towards a common goal (for example, scanning a Client spacecraft using optical, radar, and lidar sensors, or manufacturing of large space platforms). Satellite aggregation is the process of constructing platforms or spacecraft in orbit, using smaller spacecraft as the building blocks, both in a structural and a software sense [17]. With swarms of spacecraft operating in close proximity to each other, it will be essential to have a method to optimize the trajectories of each spacecraft, minimizing the risk for collisions, while allowing them to fulfill their mission operations.

3 Swarm Methodology

For the purposes of this analysis, the definition of a swarm is: a group of two or more spacecraft cooperating towards a common task or goal. The analysis is performed in the relative motion non-inertial coordinate system defined by the Clohessy-Wiltshire equations [18].

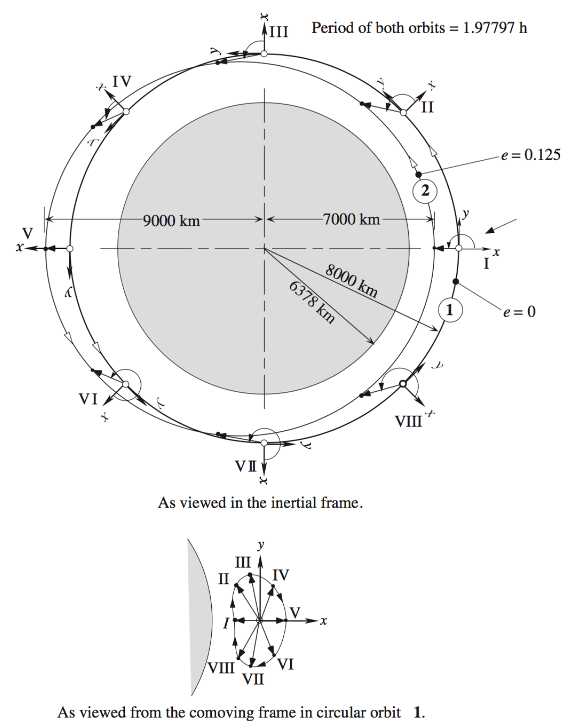

Fig. 1: Slightly eccentric orbit allows relative motion

As seen in Fig. 1, these spacecraft are in slightly different orbits from each other, such that in the relative motion space they are "orbiting" around a common point in space. The mechanics of the free-trajectory motion following these relative motion orbital tracks are well known and understood, having been used for more than fifty years, prior to the Apollo missions [5]. However, methods to autonomously maintain and guide such relative motion trajectories are not as well understood, given that robust automated rendezvous techniques have been available for just over a decade [19]. Fig. 2 shows a depiction of what a set of swarm orbits may look like in the relative motion frame.

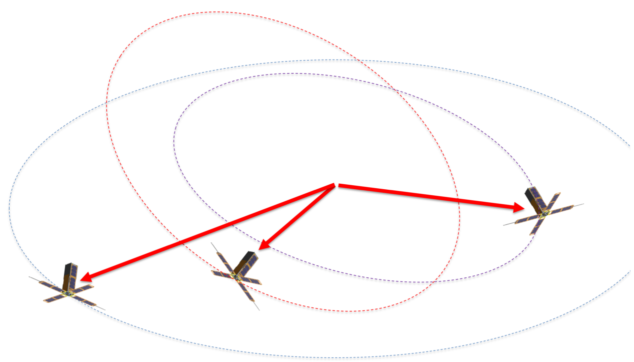

Fig. 2: Swarm of spaceraft in relative motion

3.1 Mathematical Formulation

Relative orbital motion takes place in the Local-Vertical Local-Horizontal (LVLH) rotating reference frame. This non-inertial reference frame is centered on a point in space, in orbit around the Earth, which could be a Client spacecraft, a waypoint, or some other point of interest. The x-axis is directed along the outward radial vector from the center of the Earth to the target, the z-axis is normal to the orbital plane of the target, and the y-axis lies within the orbital plane, constrained by the x- and z-axes to form a triad.

This motion can be described by the following equations of motion, where R is the vector from the center of the Earth to the Client, and δr is the vector from the Client to the Servicer vehicle:

$$\delta \ddot{r} = -\ddot{R} - \mu \frac{R+\delta r}{\|R+\delta r\|^3} \quad (1)$$

This equation of motion is a nonlinear system of equations; however, a linearized approach is desired to use in a real-time guidance application. If the target spacecraft is restricted to be in a circular orbit, the system can be defined in a closed-form linearized approximation by the Clohessy-Wiltshire (C-W) equations [18], laid out below

$$\delta \ddot{x} - 3n^2 \delta x - 2n\delta \dot{y} = 0 \quad (2)$$

$$\delta \ddot{y} + 2n\delta \dot{x} = 0 \quad (3)$$

$$\delta \ddot{z} + n^2 \delta z = 0 \quad (4)$$

These differential equations are valid while the following criterion from the linearization process holds:

$$\delta r/R << 1 \quad (5)$$

A closed form solution of these coupled partial differential equations can be obtained, expressed in matrix form below, enabling the computation of position and velocity at any point in time:

$$\delta \vec{r}(t) = [\Phi_{rr}(t)]\delta \vec{r}_0 + [\Phi_{rv}(t)]\delta \vec{v}_0 \quad (6)$$

$$\delta \vec{v}(t) = [\Phi_{vr}(t)]\delta \vec{r}_0 + [\Phi_{vv}(t)]\delta \vec{v}_0 \quad (7)$$

where the initial position and velocity are

$$\delta \vec{r}_0 = \begin{bmatrix} \delta x_0 \\ \delta y_0 \\ \delta z_0 \end{bmatrix}, \ \delta \vec{v}_0 = \begin{bmatrix} \delta u_0 \\ \delta v_0 \\ \delta w_0 \end{bmatrix}$$

n : angular rotation rate of orbit (rad/s)
t : time since initial conditions

$$\Phi_{rr}(t) = \begin{bmatrix} 4-3\cos nt & 0 & 0 \\ 6(\sin nt - nt) & 1 & 0 \\ 0 & 0 & \cos nt \end{bmatrix} \quad (8)$$

$$\Phi_{rv}(t) = \begin{bmatrix} \frac{1}{n}\sin nt & \frac{2}{n}(1-\cos nt) & 0 \\ \frac{2}{n}(\cos nt - 1) & \frac{1}{n}(4\sin nt - 3nt) & 0 \\ 0 & 0 & \frac{1}{n}\sin nt \end{bmatrix} \quad (9)$$

$$\Phi_{vr}(t) = \begin{bmatrix} 3n\sin nt & 0 & 0 \\ 6n(\cos nt - 1) & 0 & 0 \\ 0 & 0 & -n\sin nt \end{bmatrix} \tag{10}$$

$$\Phi_{vv}(t) = \begin{bmatrix} \cos nt & 2\sin nt & 0 \\ -2\sin nt & 4\cos nt - 3 & 0 \\ 0 & 0 & \cos nt \end{bmatrix} \tag{11}$$

Although the C-W equations are linearized approximations of a nonlinear system, the approximations are sufficient for the purposes of orbital rendezvous and proximity operations. The solutions diverge when the distance from the target is a significant percentage of the mean orbital radius of the target, as this is when the Earth's curvature will have an effect on the gravitational perturbations. Thus for LEO, based on the linearization criterion (Equation 5), these solutions can be used within a few dozen kilometers of the target, and in GEO within a few hundred kilometers of the target [18].

3.2 *Example RPO Trajectories*

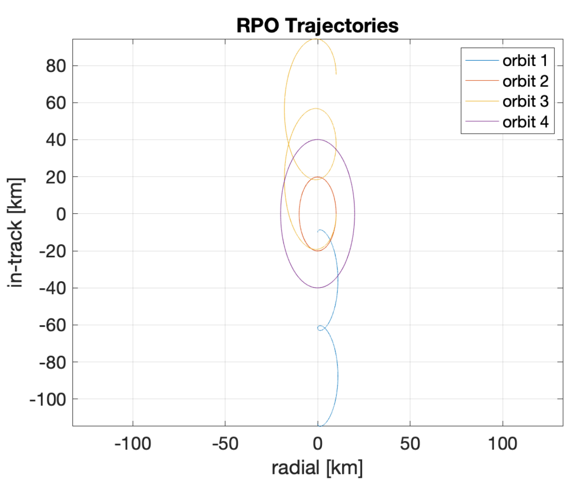

Fig. 3: RPO Trajectory Examples

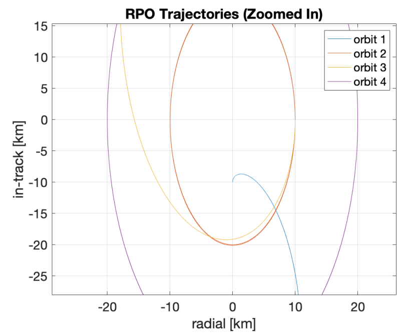

Fig. 4: RPO Trajectory Examples (Zoomed In)

Applying the C-W equations, a set of initial conditions can be propagated forwards through time to determine the resulant trajectory. Figs. 3 & 4 show an example of four trajectories, where two of them form closed "orbits" and two drift off away from the reference point towards infinity. A delicate balance between the position and velocity vectors in the LVLH frame is required to have a closed "orbit" that does not drift away, thus there are an infinite possible set of trajectories but only a small subset of those will be bound in space over time.

3.3 *Orbit Maintenance*

Now that we have defined what a relative orbit is and the trajectory that an object in relative motion will follow, the next step is to define how to maintain a relative orbit. Even if a spacecraft were injected perfectly into its orbit, there are gravitational perturbations to be considered, such as the Earth's oblateness, the Moon, and the Sun, all of which will impart tiny forces to perturb the spacecraft's orbit over time. Additionally, deviations to the planned trajectory are caused by imperfect injections into the desired orbit, leading to a drift in the trajectory compared to the nominal path.

Thus, its possible to compute the exact acceleration deviations caused by the perturbing gravitational bodies and come up with a control system to compensate for this using periodic application of thrust forces. However, for swarm RPO, this is not necessary for the most part. A rigid trajectory is generally not required, since during swarm RPO much of the focus is on entering a relative motion closed-form trajectory around a target spacecraft or body. If this trajectory deviates by a few meters, it will not affect the mission so long as all the spacecraft in the swarm are sufficiently far enough apart that a deviation of a few meters will not cause a collision (see Figs. 5 and 6). Rather than use the limited fuel resources to maintain a given trajectory, when significant deviations occur a new trajectory can be computed, which can be transitioned to while conserving propellant.

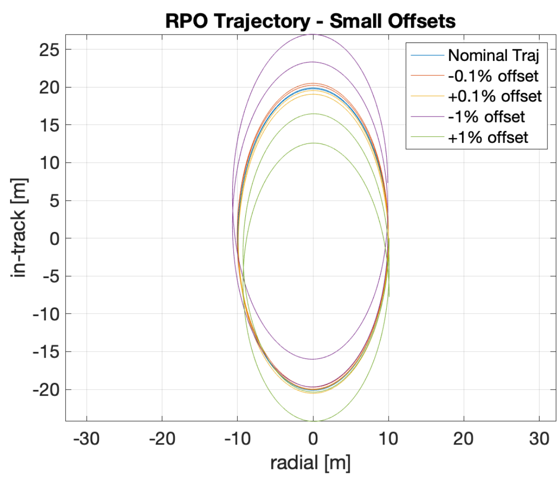

Fig. 5: Trajectory Offsets for Various Levels of Position Injection Error

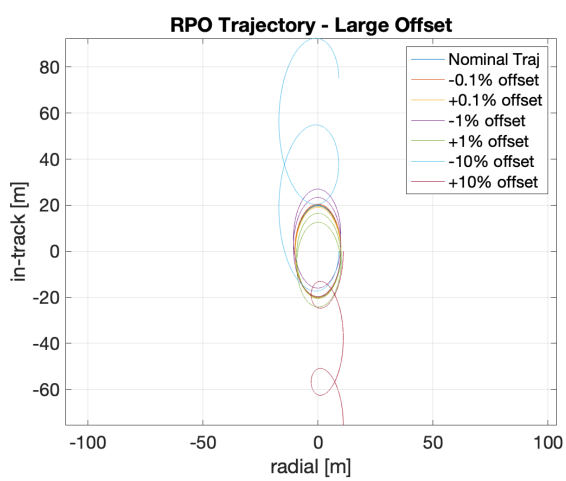

Fig. 6: Trajectory Offsets for Large Injection Errors

3.4 Perturbation Effects

In order to take into account the perturbation of the J2 effect of Earth's oblateness (the primary orbital perturbation below Geostationary orbit), a modified set of C-W equations must be derived. This mathematical problem has been solved already [20], with the equations of motion as follows:

$$\begin{aligned} x(t) = &\left(\frac{5s+3}{s-1} x_0 + \frac{2\sqrt{1+s}}{n(s-1)} \dot{y}_0 \right. \\ &\left. + \frac{1}{4} \frac{A_{J2}(3k - 2n\sqrt{1+s}) \sin^2 i}{k(-n^2 + n^2 s + 4k^2)} \right) \cos\left(nt\sqrt{1-s}\right) \\ &- \frac{1}{4} \frac{A_{J2}(3k - 2n\sqrt{1+s}) \sin^2 i}{k(-n^2 + n^2 s + 4k^2)} \cos(2kt) \\ &+ \frac{\dot{x}_0}{n\sqrt{1-s}} \sin\left(nt\sqrt{1-s}\right) - \frac{4(1+s)}{s-1} x_0 - \frac{2\sqrt{1+s}}{n(s-1)} \dot{y}_0 \end{aligned} \tag{12}$$

$$\begin{aligned} y(t) = &\left(\frac{2(5s+3)\sqrt{1+s}}{(1-s)^{3/2}} x_0 + \frac{4(1+s)}{n(1-s)^{3/2}} \dot{y}_0 \right. \\ &\left. + \frac{1}{2} \frac{A_{J2}(2ns - 3k\sqrt{1+s} + 2n) \sin^2 i}{k\sqrt{1-s}(-n^2 + n^2 s + 4k^2)} \right) \sin\left(nt\sqrt{1-s}\right) \\ &- \frac{1}{8} \frac{A_{J2}(5n^2 s + 4k^2 + 3n^2 - 6nk\sqrt{1+s}) \sin^2 i}{k^2(-n^2 + n^2 s + 4k^2)} \sin(2kt) \\ &- \frac{2\sqrt{1+s}}{n(s-1)} \dot{x}_0 \cos\left(nt\sqrt{1-s}\right) + \left(\frac{2n(5s+3)\sqrt{1+s}}{(s-1)} \dot{x}_0 \right. \\ &\left. + \frac{5s+3}{s-1} \dot{y}_0 + \frac{A_{J2} \sin^2 i}{4k} \right) t + \frac{2\sqrt{1+s}}{n(s-1)} \dot{x}_0 + y_0 \end{aligned} \tag{13}$$

$$z(t) = z_0 \cos\left(nt\sqrt{1+3s}\right) + \frac{\dot{z}}{n\sqrt{1+3s}} \sin\left(nt\sqrt{1+3s}\right) \tag{14}$$

with the terms s, c, k, and A_{J2} defined as follows:

$$s = \frac{3J_2 R_\oplus^2}{8r^2}(1 + 3\cos 2i) \tag{15}$$

$$c = \sqrt{1+s} \tag{16}$$

$$k = nc + \frac{3\sqrt{\mu} J_2 R_\oplus^2}{2\|\delta r\|^{7/2}} \cos^2 i \tag{17}$$

$$A_{J2} = -3n^2 J_2 \frac{R_\oplus^2}{\|\delta r\|} \tag{18}$$

$R_\oplus$: Mean equatorial radius of central body
J_2 : Measure of central body oblateness

Propagating a set of initial conditions using the standard linearized C-W equations (Equations 6 – 7), and the non-linear C-W equations with J2 perturbations (Equations 12 – 14), the trajectories over a period of 3 orbits can be see in Fig. 7. All trajectories in orbit will drift naturally over time, however it should be noted that when taking into account the J2 perturbation of Earth's gravity (the dominant gravitational perturbation), the direction of drift changes. This is significant for any station-keeping schema, and must be taken into account, as we will for this analysis.

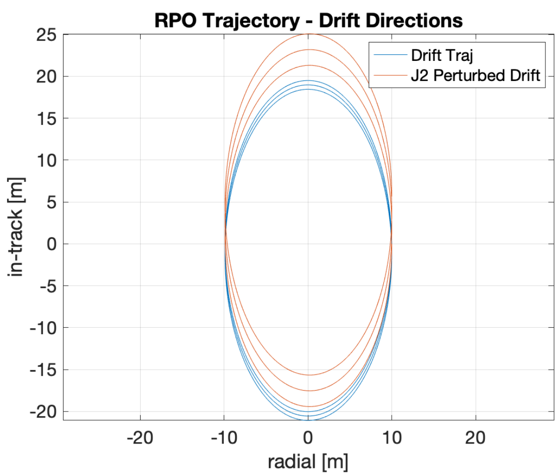

Fig. 7: Trajectory Drift for different gravity models

4 Simulated Hardware Testing Method and Results

Over the past few years, the SERC has been developing an Air Bearing Platform (ABP) testbed. This testbed is comprised of small floating platforms with pressurized air tanks that are able to use circular air bearing diaphragms to float on an air cushion over a calibrated optical glass surface. Using cold gas thrusters, these *floatbots* are able to move across the glass surface, simulating a frictionless environment in space. This makes the platform ideal for testing RPO and docking activities without the expense of a microgravity simulator such as the *vomit comet* [21], or testing aboard the International Space Station (ISS) itself [22].

Although the ultimate goal of this line of research is to implement the control algorithms outlined above on the ABP hardware testbed, the first step was to develop computer simulations of the platform. This simulated platform allowed for testing of the same control algorithms that will be loaded onto the completed ABP bots. The simulated testing uses MATLAB/Simulink coupled with NX Motion Simulation in order to model the effects of the sensors on the hardware platforms while also computing its orbital trajectory and updating its motion to match.

4.1 Simulation Details

This NX/Simulink co-simulation tool is typically used for control design, using NX Motion for system dynamics and Simulink for controller calculations. One of the benefits of this tool is that system dynamics are simulated using NX Motion, tied directly to the computer aided design (CAD) model. This minimizes the possibility of un-modeled dynamics, given that the CAD properties of the assembly as well as the forces and sensors created in NX Motion accurately represent the system; contrary to utilizing a separate mathematical model simulated by a software such as Simulink, which leaves greater room for error. An additional benefit to the co-simulation tool is the ability to animate the results for visual verification of the platform.

The co-simulation was created to design and validate ABP GNC algorithms, as well as for verification of any test planned to be run on the system. Markers, motion drivers (forces, torque, etc.), and sensors are created in NX Motion and used to to specify inputs and outputs. Generating a solution creates a Simulink NX Motion plant block, which allows integration of custom trajectories, forces and velocities into the NX simulation (in this case RPO relative motion). This enables a feedback loop which occurs for every timestep of 0.01 s. Once the simulation is completed, in addition to the values and plots determined in Simulink, it returns a result that can be animated and viewed in NX Motion.

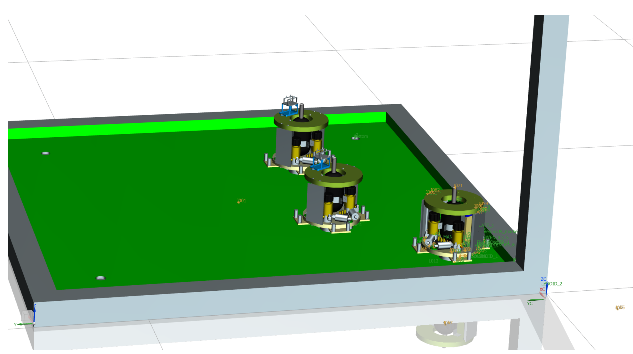

Fig. 8: Swarm of 3 floatbots on ABP

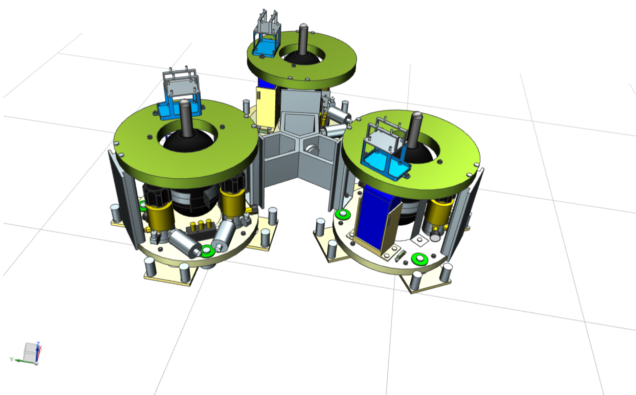

Fig. 9: Docked Swarm on ABP testbed (rendering)

4.2 *Testing Considerations*

Given that the Air Bearing Platform is a two-dimensional system with three degrees of freedom (translation in a plane, and rotation about an axis), it may seem that it iss inadequate to test rendezvous motion and control in space. However, looking at the linearized equations of relative motion (see Eq. 6), we can see that if the initial position is confined to be within the target spacecrafts orbital plane, then there is no resultant force causing any out-of-plane accelerations. This allows the ABP testbed (and its software simulation counterpart) to be used to accurately demonstrate rendezvous and docking operations in a 2D orbital plane.

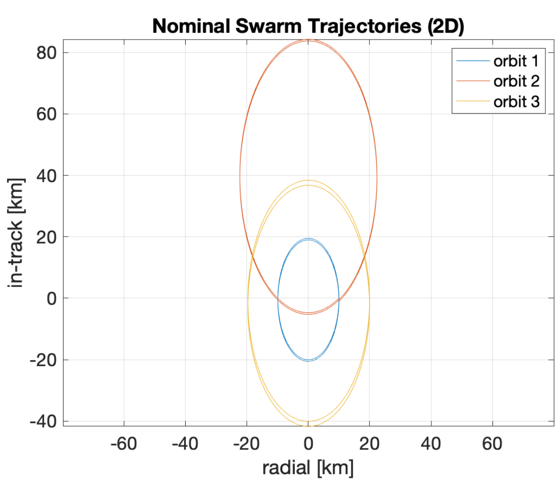

Fig. 10: Nominal Trajectories for Testbed

Before the initial testing, a set of trajectories were computed that form closed loops in the non-inertial LVLH coordinate frame. The orbit for the target spacecraft was chosen to be a circular 450 km altitude LEO orbit for simplicity. These nominal trajectories are depicted in Fig. 10. These trajectories are on the order of tens of kilometers across, and the ABP testbed is restricted to within a meter of travel in any given direction, thus the testing to be conducted on the hardware platform uses scaled values to be valid and match the limitations of the platform relative to full scale.

Following the trajectory generation, the points were fed into the ABP simulator in order to perform analysis on the sensor visibility and trajectory deviations, as well as visualization of a trajectory on a hardware system. To mimic the ABP testbed firing cold-gas thrusters to move around in a plane, the trajectory is translated into a set of thrust values to actuate the various thrusters to achieve this motion on the software testbed [23].

4.3 *Initial Trajectory Verification*

The initial simulations focused on verifying that the simulator and the input conversion programs were functioning correctly, such that the simulator could be used accurately to demonstrate the trajectories on the simulated ABP platform and be used to test aspects of swarm RPO.

Performing these initial simulations showed that it is possible to simulate swarm RPO on the ABP testbed, and allowed us to compare the resultant trajectories using cold-gas thrusters firing on the eventual air bearing table to the nominal trajectories generated mathematically in MATLAB.

4.4 *Sensor Update Testing*

To look at possible worst case we considered that future swarm based nano- and micro-satellites may not be 3-axis stabilized, and some larger spacecraft may need to be spinning to maintain thermal equilibrium and generate power using solar panels. In these cases, there may be limited windows in which the spacecraft's sensors will be able to collect data from the Client spacecraft, or communicate with other members of the swarm. Our simulation testbed was used to analyze the availability windows for a variety of spin rates and sensor Fields of View (FOV) for these possible off-nominal conditions.

Upon running these simulations, it was determined that, in accordance to our initial hypothesis, as long as the spacecraft spin-rate allows for sensor measurements of the Client spacecraft and other spacecraft in the swarm such that the time between measurements is less than the time required to drift outside of the allocated trajectory corridor at the present spacecraft velocity, then the algorithm developed for trajectory manuevering is valid.

4.5 *Trajectory Recalculation and Maneuvering*

Looking at Figs. 5 & 6, small trajectory deviations will not result in any degradation to the mission, however large offsets will. Once the difference between the nominal trajectory and the observed trajectory exceeds a specified threshold (1 km in these simulations), a new trajectory was computed, intersecting the current trajectory such that a single burn could be applied to attain it, with the new trajectory within tolerances of the nominal path while also minimizing the Δv of the burn.

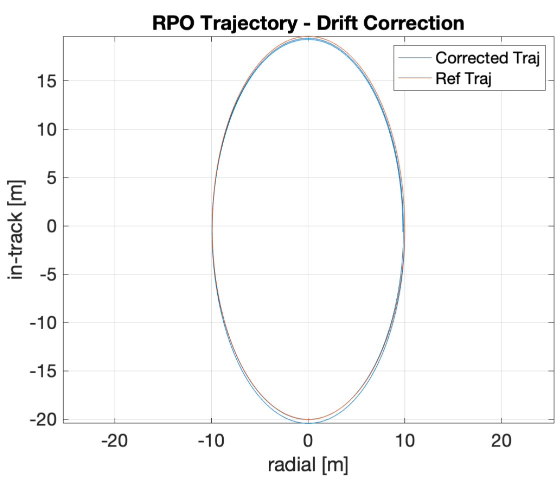

Fig. 11: Nominal (Reference) and Corrected Trajectories

Fig. 11 shows both the nominal (reference) trajectory and the corrected trajectory, corrected using periodic micro-burns to prevent the trajectory from deviating from the reference by more than 1 km. Comparing this to Fig. 7 of the free drift trajectories, this corrected trajectory maintains its position within the allocated 1 km corridor with minimal Δv application (in the example above, the Δv used for 3 orbits is 0.607 m/s). Fig. 12 below shows the deviation from the reference trajectory over time, with the peaks showing where the correction burns took place as the deviation begins to reduce.

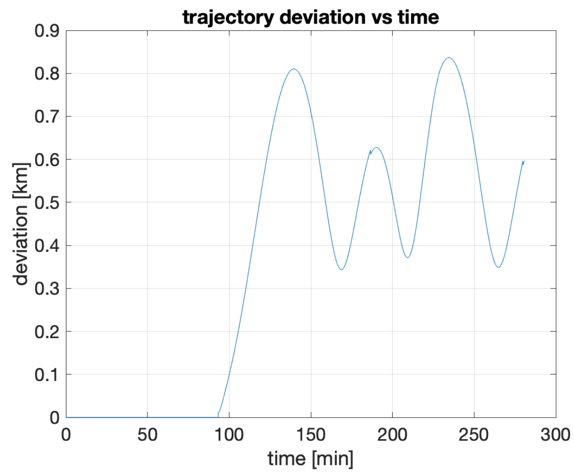

Fig. 12: Trajectory Deviation vs Time

The alternative to this dynamic method of orbital maintenance is a rigid or set orbit maintenance schema, where small corrections are made at regular intervals in order to maintain the nominal trajectory. Performing the analysis on this method for the same trajectories considered above, we found this requires an order of magnitude more Δv (in this case, the Δv jumps to 9.88 m/s for the same 3 orbit timeframe). We hypothesize that this is due to the requirement to be constantly firing the on-board thrusters to perform the large number of micro course corrections required to maintain a rigid orbital trajectory in the face of gravitational perturbations from multiple sources.

5 Future Research

Given the results from the simulation, testing the hardware and physical sensors on our planned ABP testbed will enable validation of various algorithms, enabling future simulations to be run on the MATLAB / NX Co-Sims with confidence as a step before running them on the hardware platforms. Future research will also involve testing of orbital regimes outside of LEO, including Geostationary Orbit (GEO), as well as Martian orbit for fully autonomous rendezvous operations, and around irregular gravitational bodies such as asteroids to enable swarm mining operations in space.

Additionally, the initial conditions for this simulation utilized curated trajectories that were manually determined and known *a priori* to be closed form relative motion orbits. Research is currently being conducted to create a method of optimization to solve for closed-form relative motion trajectories that can satisfy a set of mission criteria while also minimizing the collision risk of the individual members of the swarm. This optimization method is positing Genetic Algorithms as part of the optimization sequence, where full analysis and results will be presented in subsequent papers.

6 Conclusions

Based on the defined framework and the simulated ABP testbed, swarm RPO is possible demonstrating examples with small satellites that have limited fuel and resources. The simulations run show that as long as a range of trajectories are permissible over a rigid and fixed trajectory, its possible to maintain swarm trajectories for small satellites while satisfying the mission requirements for the swarm as a whole.

Future testing to transition from the simulation to the hardware ABP testbed will enable verification of the results of this analysis, and allow for expanded testing to include docking and proximity operations demonstrations.

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